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Blue Marble: Remote Characterization of Habitable Planets

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ABSTRACT

The study of the nature and distribution of habitable environments beyond the Solar System is a key area for Astrobiology research. At the present time, our Earth is the only habitable planet that can be characterized in the same way that we might characterize planets beyond the Solar System. Due to limitations in our current and near-future technology, it is likely that extra-solar planets will be observed as single-pixel objects. To understand this data, we must develop skills in analyzing and interpreting the radiation obtained from a single pixel. These skills must include the study of the time variation of the radiation, and the range of its photometric, spectroscopic and polarimetric properties. In addition, to understand whether we are properly analyzing the single pixel data, we need to compare it with a ground truth of modest resolution images in key spectral bands. This paper discusses the concept for a mission called Blue Marble that would obtain data of the Earth using a combination of spectropolarimetry, spectrophotometry, and selected band imaging. To obtain imagery of the proper resolution, it is desirable to place the Blue Marble spacecraft no closer than the outer region of cis-lunar space. This paper explores a conceptual mission design that takes advantage of low-cost launchers, bus designs and mission elements to provide a cost effective observing platform located at one of the stable Earth-moon Lagrangian points (L_4, L_5). The mission design allows for the development and use of novel technologies, such as a spinning moon sensor for attitude control, and leverages lessons-learned from previous low-cost spacecraft such as Lunar Prospector to yield a low-risk mission concept.

INTRODUCTION

The study of the nature and distribution of habitable environments beyond the Solar System is a key area of Astrobiology research. The first goal of NASA's Astrobiology Roadmap is to understand the nature and distribution of habitable environments in the Universe. It directs us to "conduct astronomical, theoretical, and laboratory spectroscopic investigations to support planning for and interpretation of data from missions designed to detect and characterize extrasolar planets.[1]"

Initial data returned from missions such as Terrestrial Planet Finder (TPF) are not likely to provide data that requires extensive analysis and modeling. On the other hand, future missions will provide superior instruments that will provide additional data for exploitation. However, not only do we not currently know how to analyze such superior data, we do not even know what data might be most valuable. The principle goal of the Blue Marble mission is to develop sufficient understanding of the "observables" of Earth to support planning observations of extrasolar terrestrial planets and development of observing instruments for future platforms.

One definite feature of near future observational programs is that these planets will be viewed as single pixel objects. The requirements for imaging extrasolar planets are beyond our current and likely near-future technologies. As such, skills and processes in analyzing and interpreting the radiation from single pixel objects must be developed and honed. These skills include the study of the time variation of the radiation and the

range of its photometric, spectroscopic and polarimetric properties.

To properly interpret and analyze single pixel data, it needs to be compared with a ground truth of modest resolution images in key spectral bands. The key data to be returned from Blue Marble is a combination of spectropolarimetry, spectrophotometry and selected band imaging.

The whole of Earth is not seen from nearby space. A complete hemispherical view of Earth can only be obtained from an infinite distance. At closer distances the region not seen past the limb increases with the observed planet angular diameter. To obtain data of use for proper study, Blue Marble should observe the Earth from a distance no closer than the outer region of cis-lunar space. Blue Marble is therefore architected to be a mission that could be placed at any of the Earth-Moon Lagrangian points or in lunar orbit.

SCIENCE

Science Background

Early research into the detection and characterization of habitable planets suggested that the mid-infrared spectral region would be useful as that spectrum showed the presence of water, ozone and CO_2[2]. Comparisons of the mid-IR of planets obtained using "Nimbus" Fourier Transform Spectrometers helped confirm the selection of this observing band[3,4]. Subsequently, this region was selected as the basis for a TPF space mission[5].

Later studies indicated that the visible and near-IR spectral regions show different molecular features of interest[1]. The combination of observations from 0.3 - 25 µm spectral bands yields more detailed information as the mid-IR spectra provide information about temperature and atmospheric vertical thermal structure while the visible and near-IR spectra give better indicators of molecular quantities. In addition, molecules of interest in determining planet habitability provide different response in visible and IR frequencies.

Additional observations were made of the visible spectrum of the Earth using lunar Earthshine[6]. When compared to predictive models, the observations showed additional features such as strong Rayleigh scattering in the blue spectral region, the vegetation "red edge" indicating the presence of terrestrial vegetation, and an additional feature later identified as O_4 which has use as an indicator of atmospheric pressure. The spectra showed a red-edge difference from when the moon was viewing oceans and when the moon was illuminated by light reflected from the Amazon basin. These Earthshine observations did not include polarization or illumination phase variation as Earthshine is only observable near the new moon. Additional observations of Earthshine were extended into the near-IR, but thermal radiation from the moon placed an upper limit of ~2.4 µm for useful data[7].

These Earthshine observations have prompted us to consider what other new information might be available by observing the whole Earth. Neither the linear nor circular polarization of the whole Earth has been observed. There are possibilities of observing the polarization effect of the glint of the sun on the oceans or from cloud polarization. Both of these are expected to show Brewster angle effects with a peak polarization that can identify the refractive index of the cloud / liquid material. These are the only currently known methods to detect liquid water. The presence of atmospheric H_2O gas bands is no indication of the presence of the liquid phase and it is the liquid phase that is crucial to terrestrial life. The necessary observations to make are the variation of polarization with illumination phase angle[8].

Laboratory measurements of the reflected light at the peak of the chlorophyll a band near 6600A yields optical circular polarization (Q) of 0.1%. This result indicates the level of sensitivity and coverage that would allow remote detection. The best way to prepare might be to make a survey of the circular polarization signatures of planet Earth[9]. Wolstencroft et al. suggested that "The global polarization and reflectance properties of Planet Earth have been measured by the POLDER satellite: at 443nm atmospheric Rayleigh scattering dominates, but at 865nm the average surface properties of ocean, vegetation, desert and snow can be estimated. For cloud-free surfaces at 865nm and 90 degree phase angle the percentage polarization, p, and reflectance, R, are respectively [55%, 9%] (ocean), [7%, 23%] (vegetation), [6%, 40%] (desert) and [3%, 80%] (snow). Note that the values for clear and cloudy ocean are very different, viz [55%, 9%] and [4%, 45%] respectively. Allowing for the fractional global areas of each component and a global cloud cover of 55% yields p=7.3% for a pale-blue-dot Earth. pR is greatest for oceans and least for vegetation and hence the prospects for detecting pR from vegetation on an Earth-like planet are poor unless >50% is covered in vegetation. However prospects of using the phase angle and wavelength dependence of a pale-blue-dot planet to deduce its properties as it rotates and orbits are more encouraging: the main obstacle will be to overcome the difficulty of correcting for the unknown and variable cloud cover. The possible application of a circular polarization signal that is unique to vegetation remains

an intriguing possibility for remote sensing that requires further study.[10]"

From these references, it is clear that the characteristics of Earth as an example of a single pixel object in motion around a star are not yet adequately determined. Seager et al. write "A time series of photometric data of a spatially unresolved planet could reveal a wealth of information such as the existence of weather, the planet's rotation rate, the presence of large oceans of surface ice, and the existence of seasons.[11]" To make such observations and test the validity of the process, we need to observe one object where we can obtain both spectrophotometric data and spectropolarimetric data as well ground truth images across appropriate wavelengths.

Science Goals

The primary science goals of the Blue Marble mission concept as well as science measurements, instruments and mission impacts are summarized in Table 1.

MISSION DESIGN

Three primary requirements for the Blue Marble mission design are obtained from the science goals as well as a fourth requirement dictated by the instrument implementation.

Table 1: Blue Marble Mission Requirements

Science Objective	Science Measurement	Instrument	Instrument Requirements	Mission Requirements
Obtain linear spectroscopy of full Earth Obtain circular spectroscopy of full Earth	• Observe at least 98% of earth disc • Observe in VNIR wavelengths • Observe full 360° phase angle • Observe full range of Earth seasons	Spectropolarimeter / Polarizing Imager	• Rotate instrument about boresight at 0.5 to 1 rpm • Collect 8 - 12 samples per spacecraft rotation • Instrument bandpass from 350 to 900 nm	• Mission orbit apoapse > 200,000 km • Orient spin axis to <15 arc-sec error • Observe Earth in 360° angle per month • Mission lifetime > 1 year
Obtain polarized imagery of full earth	• Observe at 4 bands		• Observe 400, 570, 650, and 760 nm • Resolution of at least 400 x 400 pixels • FPA readout rate of 5 to 20 Hz	
Obtain infrared spectroscopy of full earth	• Observe in M/LWIR band	IR Spectrometer	• Instrument bandpass from 5 to 20 microns • <125 nm resolution	• Mission orbit apoapse > 200,000 km
Obtain infrared imagery of full earth	• Observe in M/LWIR band	IR Imager	• Resolution of at least 128 x 128 pixels • Imager bandpass from 5 to 20 microns • FPA readout rate of 5 to 20 Hz	• Mission orbit apoapse > 200,000 km
Obtain ground truth imagery	• Observe in 3 colors	Visible Camera	• Instrument bandpass from 350 to 900 nm	• Mission orbit apoapse > 200,000 km

Primary Objectives
Secondary Objectives
Tertiary Objective

Mission Destination

First, in order to observe at least 98% of the Earth's hemisphere, Blue Marble needs an operational orbit with an apoapse altitude of at least 200,000 km. Potential mission orbits could be similar to those used for IBEX, THEMIS, and Lunar Prospector thus showing feasibility for low cost missions at such altitudes. The desire to acquire data over the full Earth over a wide range of phase angles and the desire to standardize mission operations leads to further investigation.

Instruments at lunar surface locations could be used, but would suffer from 14 day unavailability of power each lunation and are at risk of effects from lunar dust and thermal extremes. Polar lunar orbits that have precession rates matching that of the Earth-Moon system could be used, but close proximity to the Moon leads to frequent solar power outages.

Alternative observing stations exist at the Lagrangian points of the Earth-moon system. As shown in Figure 1, five Lagrangian points exist; of which three, L_1, L_2, and L_3 are unstable. Spacecraft can be stationkept in halo or Lissajous orbits around these points but require active propulsion to maintain their orbits. The L_4 and L_5 points may be more useful since they are stable to first order, although solar influence and n-body effects from the planets will perturb the mission orbit [12]. In addition to these effects, the possibility of debris impacts near the points themselves is of concern and thus a similar Lissajous orbit would be used. Further study on whether one of the stable or unstable points is optimum should be conducted, for purposes of this paper; a mission to one of the two triangular points will be selected.

Mission Lifetime

The second mission design requirement imposed deals with the required mission lifetime. Since significant variance exists in the polarization and reflectance values dependent upon water phase and cloud cover, it is desirable to take measurements over a wide range of seasonal weather conditions. This goal leads to a nominal mission lifetime of at least one year.

The third design requirement levied by the science goals is to collect polarimetric and imaging data over a full range of phase angles. In addition, it is desirable to obtain phase angle samples during major seasons. As such the full range of phase angles should be traversed at least every 90 days. This requirement effectively excludes missions located at the Sun-Earth system Lagrangian points as the phase angles will remain nearly constant. Lunar orbits or lunar Lagrangian point orbits will see the full range of phase angles in approximately 29 days, allowing for full phase angle sampling during all seasons.

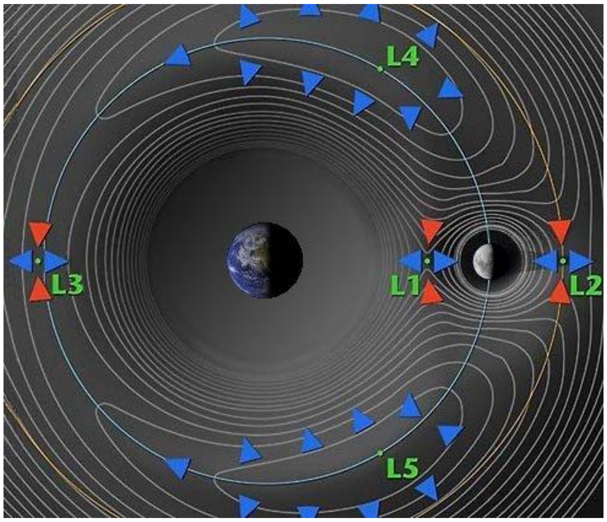

Figure 1: Lagrangian Points in the Earth-Moon System

Space Vehicle Attitude

Finally, a derived requirement relates to the stabilization of the spacecraft. In order to properly measure polarization, either the focal plane or aperture grating must be rotated. To simplify instrument designs and vehicle operations, spinning the observing spacecraft becomes an attractive solution.

By spinning the space vehicle, intrinsic instrument polarization is rotated about the polarization of the observed object and can be cancelled out. Circular polarizations are converted to linear for measurement and any linear leaks are modulated by the rotation and can be extracted from the data.

While full spacecraft rotation serves to stabilize the platform attitude, there are disadvantages to this method. Since the spacecraft is rotating about the Earth in its ~27 day orbit, a point on the Earth's surface will be moving under the spacecraft rotation axis at a rate of ~30 km/minute. However, the Earth's equatorial regions are simultaneously moving at a similar speed around the Earth's axis. Therefore, the allowable exposure time before features exhibit unacceptable blur is similar from both causes. To continue imaging the Earth, the spacecraft's spin axis will need to precess over the 27 day orbit duration. The method for accomplishing this rotation will be discussed in the Spacecraft Design section of this paper.

The rotation rate for the spacecraft is set by requiring that after one rotation, the image studied for polarization ground truth will not have changed by an amount that disturbs the measurement of polarization. This condition is most severe for points on the equator near the disk center. Thus since the rotation of the Earth there is 30 km/minute, if the vehicle rotation rate is 1 rpm, pixels cannot be smaller than 30 km. Similarly, at a rotation rate of 0.5 rpm, the pixels should be larger than 60 km. These rotation rates correspond with 400 and 200 pixels polarization images respectively. A lower bound on space vehicle rotation rate can be set by noting that 200 pixel images essentially limit useful data return.

An upper bound on rotation rate can correspondingly be set by noting that pixels at the Earth limb will be similarly blurred. For a 1 rpm rotation rate, 30 km pixels at the Earth limb will exhibit blurring for exposure times greater than 1/20th of a second. At 0.5 rpm, 60 km pixels will blur for exposures in excess of 1/5th of a second.

For spectropolarimetry, 8 to 12 observations per spacecraft rotation are needed to properly measure polarization. At a rotation rate of 1 rpm, 5 seconds are available per observation including exposure time and detector readout. Similar instruments making measurements from the ground perform observations requiring less than 1/10th of a second showing compatibility with requirements.

The required pointing precision for the vehicle spin axis can be set by examining the tolerable error within the size of camera images. For a given number of pixels spread across the Earth disk of 2°, the number of pixels in the detector must be increased by a factor of 1+x where x is the pointing error in degrees. To limit the increase in pixel count to no more than 25%, the maximum allowable error in pointing axis orientation is 15 arc-minutes. This error is similar in magnitude to that which will build up in ½ hour without precession of the vehicle spin axis.

In conclusion, a vehicle spin rate between 0.5 and 1 rpm seems to be optimum in terms of minimizing error and maintaining compatibility with current instrumentation.

Mission Orbit Transfer

Placing Blue Marble at one of the triangular Lagrangian points can be expensive in terms of injection energy or expendable propellant required[13]. The direct, traditional method of transfer uses a Hohmann transfer with the launch vehicle delivering the spacecraft into an orbit with periapse altitude of 200 km and apoapse at lunar distance. An equivalent method is to calculate the ΔV required to transfer from a circular parking orbit. Using a 200 km orbit at 28.5° inclination, the departure ΔV can be calculated to be 3135 m/s. Rendezvous ΔV can be calculated to be 830 m/s.

An alternative method to transit to the Lagrangian points would be to use a lunar flyby to change the energy of the Earth-relative orbit. Slightly higher departure ΔV will be required as the lunar flyby will not take place at the apoapse of the transfer orbit and is expected to range from 3150 to 3180 m/s. Rendezvous ΔV will vary depending on the flyby trajectory designed and should vary from 400 to 700 m/s.

An innovative method to minimize total required ΔV is to use a Weak Stability Boundary (WSB) transfer trajectory such as that used for Hiten[14]. In this case, the spacecraft is injected into a highly elliptical orbit with apoapse ~1,500,000 km from the Earth. The orbit apoapse is oriented such that solar perturbations cause the spacecraft to fly by the Moon into an orbit similar to that of the Moon with a slightly different period. This orbit will periodically pass the triangular Lagrangian points at which time a small rendezvous ΔV will place the spacecraft into its Lissajous orbit. The departure ΔV required will be ~3200 m/s and the rendezvous ΔV will be ~370 m/s depending upon the conditions of the final Lissajous orbit.

This reduction in total ΔV comes at the cost of transfer time. All three transfer methods will require small amounts of propellant for trajectory correction maneuvers. These maneuvers should be small and 50 m/s has been budgeted for all three transfers. A summary of these transfer methods is shown in Table 2. Further study of the WSB trajectory is required, but this option has been selected for this paper.

Table 2: Summary of Transfer ΔV Requirements

	Direct	Lunar Flyby	WSB
Injection ΔV (m/s)	3135	3180	3200
TCM ΔV (m/s)	50	50	50
Rendezvous ΔV (m/s)	830	700	370
Total ΔV (m/s)	3965	3880	3570
Transfer Time (days)	5	~8	~100

Orbit Stationkeeping

Orbits around L_4 and L_5 are defined by the natural dynamics of a quasi-stable location and can remain stable for extended periods. Motion about these points is determined by fundamental frequencies that define short and long period behavior. These frequencies are influenced by the Earth-Moon mass ratio and solar gravity perturbation[15]. A method for estimating the fuel required can be constructed by placing the spacecraft on a point on the Earth-L_4 (or Earth-L_5) line with velocity perpendicular to the line. One can then constrain the C3 energy of the orbit with respect to the Earth. Setting the C3 limit to -1.00 km^2/s^2 keeps the spacecraft within orbit about the L_4 or L_5 point for several years. Detailed numerical simulations are required to correctly estimate the stationkeeping ΔV required, however an upper limit for Blue Marble can be estimated from simulations of spacecraft larger than Blue Marble. Such estimates indicate the need for approximately 30 m/s/year for small spacecraft in a range of Lissajous orbits around the triangular points[16]. Detailed modeling will be required for full system design.

INSTRUMENTATION

For the Blue Marble mission concept, it is assumed that the spectrophotometrics and imaging unpolarized observations of Earth in the 300 to 900 nm spectral bands will be extracted from appropriately averaging polarized observations. This instrument will require a design to match the characteristics of the platform.

The currently preferred instrument concept is to use a slit entrance aperture followed by a field lens which images the Earth onto a grating. The slit would have grating order-sorting filters that divide its useful length in two. Prior to light entering the grating, it would pass through a Wollaston prism which would divide the two polarizations along the direction of the slit with a 2.5° separation.

A camera would then image the resultant spectrum onto a CCD. Four spectra would then be lined up on each other; first order (red), polarization parallel, second order (blue), first order polarization perpendicular and second order polarization perpendicular. As the spacecraft rotates, the intensity of the perpendicular and parallel polarizations will be modulated by a sine wave antiphase to each other. These signals will produce two results for the percentage linear polarization and position angle and the average results will be a measurement at each spectral point. The use of two channels will remove most systematic errors in measurement.

For Blue Marble, the instrument would have an entrance focal length of 30 cm, a ~2 cm grating, and a 10 cm focal length camera. The entrance slit width would be ~3 times the CCD pixel diameter and around 2 cm long. The grating would be a reflection grating with 300 grooves/mm blazed around 800 nm first order. The readout CCD is assumed to be a 512 x 512 pixel device.

Imaging functions would be accomplished by placing a similar Wollaston prism in the beam between a camera and the Earth. For 200 pixel images of Earth the camera would have a 6 cm focal length and focal ratio of ~F/5. The lens aperture would be approximately 1 cm in diameter.

Analogies to instruments with similar resolution, focal length and aperture diameter yield an instrument with an estimated mass of 10 kg and readout power draw of 5 W.

Larger optics are required for the secondary payloads on the mission. The first is a multi-bolometer array covering at least the 5 to 20 micron band with resolution of around 512 x 512 pixels. Greater range is desired if possible, i.e. 2.5 to 40 microns. Similar instruments based off the Boeing U3000 uncooled microbolometer suggest an instrument with a mass of around 5.5 kg and an active power of 6 W.

Second, a M/LWIR spectrometer that covers similar bands with a resolution of around 125 nm is desired. The Mini-TES instrument on MER provides similar performance and serves as an analogous instrument with mass around 3 kg and readout power of 6 W.

Finally, a three band visible camera with megapixel resolution could provide images of Earth for public consumption. Apollo images of Earth were very popular with the American public. High resolution images of the Earth from a variety of phase angles would be expected to engage the public interest as well as provide truth knowledge in visible wavelengths. A 1 megapixel image would require a camera capable of 1/100th of a second and an F/5 camera with a lens aperture of ~6 cm would be appropriate. Analogies to similar cameras yield a mass estimate of 3 kg and a power draw of around 2 W. A summary of the instruments mass and power including growth contingency is shown in Table 3.

Table 3: Blue Marble Payload Summary

Name	Mass (kg)	Power (W)
Spectropolarimeter / Polarizing Image	10.0	6.3
IR Imager	4.4	6.6
IR Spectrometer	3.3	6.6
Visible Imager	3.8	2.5
Total	**21.5**	**22.0**

SPACECRAFT

The Blue Marble spacecraft has been implemented as a simple spacecraft drawing heritage from the NASA Ames Research Center CommonBus design. Designed to operate in spinning modes at trans-lunar distances, the CommonBus platform will easily provide the required mission lifetime and services for Blue Marble. Blue Marble in its deployed configuration is shown in Figure 2.

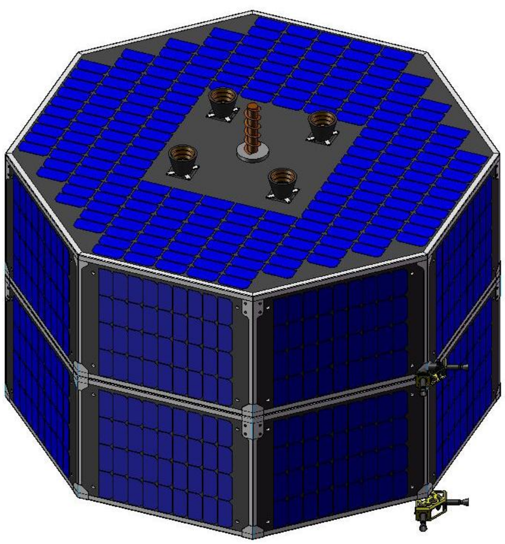

Figure 2: Blue Marble Deployed Configuration

ADCS

While spin stabilization has been used for many years, spacecraft using this technique are typically viewing stationary inertial targets. For spinning spacecraft that track targets on the Earth, such as geo-communications spacecraft, a despin platform is traditionally employed to allow the communications payload to remain pointed at the ground while the spacecraft bus spins. While this design may work for Blue Marble, the cost and complexity associated with such platforms typically restricts their use on small spacecraft.

Rather than use these methods, Blue Marble proposes to slowly precess its spin axis to match the Earth's rotation. At least two methods for accomplishing this precession are available. First, the spacecraft could apply small torques perpendicular to the orbit place while imaging is not taking place causing the spin axis to precess to track the Earth. The required torque to be applied can be calculated as $d/dt(I\omega)$ where I is the moment of inertia in question and ω is the vehicle rotation rate. The slow angular rates required by Blue Marble help to minimize fuel usage.

While the fuel usage would be small, the Blue Marble thrusters would be rather inefficient working in a low duty cycle pulsed mode. Another concern is with fuel sloshing due to the applied torques while spinning. While initial calculations show the coning angle of the spacecraft to be within pointing requirements, higher fidelity modeling should be performed to confirm the viability of this control technique as well as projected fuel usage.

A second control method for accomplishing spin axis precession would be to spin the vehicle for observations and then despin it to move the axis. This control method would be viable but over the expected mission lifetime would consume around 24 kg of propellant, more than can be provided in the CommonBus baseline configuration[17]. Another option to implement this control method would be to use a single large reaction wheel mounted on the spacecraft spin axis that would perform both spin and despin maneuvers while reorientation maneuvers would be performed by small thrusters. Current estimates show that an 8 kg reaction wheel using around 20 W of power would be sufficient and that around 3 kg of hydrazine would be required for spin axis reorientation.

For attitude reference, six coarse sun sensors have been selected for launch and emergency modes. A small star tracker providing around 1 arc-min accuracy (3σ) yields primary attitude knowledge with three small single axis micro-machined quartz rate sensors providing propagation knowledge. The quartz rate sensors may not be required as the maximum spin rate of 1 rpm corresponds with a 6°/sec rotation rate that some comparable star trackers can meet. Further testing and simulation may alleviate the need for these sensors. A summary of the mass and power allocated for the ADCS subsystem including growth contingency is shown in Table 4.

Table 4: ADCS Subsystem Summary

Component	Number	Total Mass (kg)	Total Power (W)
Reaction Wheel	1	8.8	16.8
Coarse Sun Sensors	6	0.1	0.0
Star Tracker	1	0.6	2.1
Rate Gyros	3	0.3	2.4
Total		**9.8**	**21.3**

The proposed sensor suite should provide around 1.2 arc-minutes knowledge enabling control of the spin axis to around 4.5 – 5.0 arc-minutes. As such 10 arc-minutes of error, corresponding with around 20 minutes of observation time are available for each pointing. This would allow around 4 sets of observations to be collected at each location with 24 observations available per day.

While not required for Blue Marble an interesting possibility exists for testing new technology that could support future lunar missions. Several mission possibilities to the triangular Lagrangian points have been proposed. As this point will remain very close to 60° from the Moon, an Earth or Sun sensor could be modified to provide spacecraft attitude knowledge. A spinning Moon sensor could be used to support alternative missions that require spin rates that are incompatible while a fixed aspect sensor could provide a low cost method for providing ancillary information. As the current mission design provides both spinning and fixed mission segments, both types of sensor or a joint sensor could be tested for future use.

Command and Data Handling

The processing and data collection rates for Blue Marble are fairly modest. Space vehicle processing is limited to control functions and vehicle health while also providing data storage and processing for the payloads. A summary of the data acquisition rates and data acquisition volumes are shown in Table 5.

Table 5: C&DH Data Handling Requirements

	# / Images / Observation	Data Volume (Mbits)	Data Rate (Mbps)
Spectropolarimeter	12	0.09	0.02
Polarizing Imager	1	3.67	0.73
IR Imager	1	0.23	0.05
IR Spectrometer	1	0.01	0.01
Visible Imager	1	16.78	3.36

These requirements are set for 14 bit depth data sets for each measurement and assume a lossless data compression ratio of 2 to 1 prior to storage in memory. With 96 sets of these observations in a 24 hour period, the daily data volume of science data can be calculated to be 124 MB setting a lower limit on allowable data volume. These data rates and volumes are compatible with the Broad Reach Engineering Integrated Avionics Unit selected for the ARC CommonBus. The IAU also handles solar array power and charging control for the space vehicle.

Communications

While the space vehicle must communicate with Earth from lunar distance, this link can be accomplished without requiring the use of the Deep Space Network, allowing more cost effective operations. Since the top face of Blue Marble will be aligned with Earth at all times, a modestly directional antenna can be used for primary communications with smaller nearly omni-directional patches being used for launch and emergency mode communications.

To provide bandwidth efficient communications, a small S/X-Band transponder has been selected. Downlink communications have been implemented as a BPSK link at 64 kbps in X-band with a 4 kbps S-band uplink from the ground. A small helix antenna similar to that used for Genesis provides around 16 dB gain which closes both links with margin. The 64 kbps downlink allows the use of USN assets or equivalent for around 5 hours per day. Further optimization of this link is possible. A summary of the communications equipment including growth contingency is shown in Table 6.

Table 6: Communications Subsystem Summary

Component	Number	Total Mass (kg)	Total Power (W)
S/X Transponder	1	1.0	30.0
Helical MGA	1	0.6	0
Patch LGA	2	0.2	0
Filter	1	0.6	0
Triplexer	1	0.6	0
Switches, Misc	2	1.7	0
Coax		1.3	0
Total		**5.2**	**33.0**

Despite the simple nature of the communications system, it is able to provide sufficient power. The use of X-band for the downlink moves away from the crowded portions of the spectrum to facilitate easier frequency registration. If further study warrants the link could be moved entirely to the X-band. A summary of the communications system link budget in nominal modes is shown in Table 7.

Table 7: Link Budget Summary

	X-Band D/L	S-Band U/L
Contacts / Day	4	
Time / Contact	75 mins	
Frequency	8.15 GHz	2.1 GHz
Data Rate	64 kbps	4 kbps
EIRP	22.1 dB-W	78 dB-W
SC Antenna	MGA Helix	LGA Omni
Antenna Gain	16.2 dB	0 dB
Modulation	BPSK	BPSK
Encoding	None	None
BER	1E-5	1E-5
Required Eb/No	9.6 dB	9.6 dB
Margin	4.6 dB	40.3 dB

Power

The ARC CommonBus platform uses body mounted solar arrays to minimize cost and risk associated with deployment. Using 28% efficient triple junction cells, the platform provides around 150 W at BOL. All eight octagonal faces of the side panels are populated with cells, as are parts of the top and bottom faces. Required power generation at BOL is 109 W including contingency yielding margin on the arrays of 38%. If greater margin is desired, higher efficiency cells or a slightly extended array can be added.

To provide power during launch modes and emergency modes a small 13.5 A-hr lithium ion battery has been specified. With no regular eclipses, the cycle life on the battery will be extremely low and battery DoD can be extended to 90 – 95% on the few occasions that require its use. Solar array string switching and battery charging is controlled through the C&DH system. Most components on the bus do not require power regulation, but mass has been allocated to condition and regulate power for the payload components. A summary of the components selected for the power subsystem including growth contingency is shown in Table 8.

Table 8: Power Subsystem Summary

Component	Number	Total Mass (kg)
Solar Arrays		2.2
Battery	1	3.9
PCAD	1	1.5
Harness	1	5.6
Total		**13.2**

All other propulsive maneuvers would be performed by a small dual mode bipropellant system. Injection into the Blue Marble mission orbit and periodic stationkeeping would be performed using the primary bipropellant system. Spin axis precession and reaction wheel unloading would be accomplished using eight small hydrazine thrusters. A small amount of ΔV is allocated for Blue Marble to depart the L_4 / L_5 vicinity at end of mission. A summary of the ΔV maneuvers required and propellant allocated is shown in Table 9.

Table 9: Propellant Summary

Maneuver	ΔV (m/s)	Propellant Mass (kg)
Departure	3200	319.8
TCMs / Insertion	420	14.5
Spin Axis Precession	70	3.2
Stationkeeping	60	1.9
Deorbit	30	1.3
Total	**3780**	**20.9**

Structures

The structural system for Blue Marble is designed to use lightweight carbon composite panels with aluminum fittings and longerons. The design provides high stiffness and strength while providing a mass fraction of around 18% of vehicle dry mass.

Thermal

Due to the spinning spacecraft attitude, it is possible to use passive thermal control techniques as the attitude tends to even temperatures across the vehicle. Mass has been allocated for heat strapping if necessary on the top and bottom panels. Heater power for nominal and survival modes is available for all components.

Summary

As described, the Blue Marble design uses simple designs and previous NASA and Air Force investments to produce a low mass, low risk spacecraft. A summary of the Blue Marble free flyer mass and power budgets is shown in Table 10. Growth contingency is summed separately in this table.

Table 10: Blue Marble System Summary

Subsystem	Mass (kg)	Power (W)
Payload	18.0	4.8
ADCS	8.9	20.4
C&DH	5.2	35.0
Communications	5.2	10.0
Power	11.4	15.0
Propulsion	16.2	1.0
Structure	14.8	
Thermal	2.4	4.5
Growth	11.3	5.0
Propellant	22.2	
S/C Adaptor / Half	3.0	
Total Wet	**118.6**	**95.7**

As described, in addition to the Blue Marble spacecraft, an SRM and associated support structure is required for the current mission implementation. Simple metallic structures and heritage separation systems requiring no new development work have been selected for use. A summary of the masses of the Blue Marble launch stack are shown in Table 12.

Table 11: Blue Marble Launch Mass Summary

Module	Mass (kg)
Blue Marble	118.6
S/C – SRM Adaptor / Half	7.0
SRM	344.1
SRM to LV Adaptor Skirt	29.5
LV Adaptor	10.0
Total	**509.2**

LAUNCH

As a small, low cost mission, Blue Marble also needs to be paired with a small low cost launch vehicle to make the entire mission cost effective. Three vehicles are available in the cost and performance class to deliver Blue Marble to orbit. As stated before, the mission ΔV requirements shown in Table 2 are calculated from a 185 km circular by 28.5° inclination orbit. Five vehicles were considered for Blue Marble, the Pegasus XL, Minotaur I and Minotaur IV from Orbital Sciences and the Falcon 1 and Falcon 1E from Space Exploration Technologies. The Falcon 1 and Pegasus XL have insufficient lift capacity to the target orbit. A summary of the performance and margin available from the remaining three vehicles is shown in Table 12.

Table 12: Launch Vehicle Summary

Vehicle	Launch Capacity (kg)	Margin (%)
Falcon 1E	700	37.5%
Minotaur I	575	12.9%
Minotaur IV	1700	233.9%

While the Minotaur I shows positive margin, the margin available is less than required in GSFC-STD-1000 "Gold Rules". The Falcon 1E has not successfully flown but is currently on schedule for a CY2010 launch, which would be compatible with the required development schedule for Blue Marble. This paper considers selecting the Falcon 1E as the launch vehicle to be acceptable with the availability of the Minotaur IV as a backup option. Blue Marble can be seen in its launch configuration with the Falcon 1E fairing in Figure 3. As shown, no major concerns with clearances within the fairing exist.

CONCLUSION

The Blue Marble mission concept as presented in this paper shows a feasible means to use a small spacecraft which will obtain key data to guide astrobiology research and future instrument design. The mission design proposed builds upon previous NASA missions in such a way that lessons learned in their design will be directly applicable to Blue Marble.

Using a combination of a small onboard propulsion system with an SRM allows transfer to one of the triangular Lagrangian points in the Earth-Moon system. Viewing the Earth from this location enables hemispherical views of the Earth under a wide range of phase angles and seasonal variation. The selection of spectropolarimetric, spectrophotometric and imaging data in select bands will provide the data required for scientists to focus their research in key bands that indicate the presence of liquid water phases and possible vegetation. Using small spacecraft, low cost launch vehicles and commercially available ground system networks allows Blue Marble to be implemented in a cost effective manner.

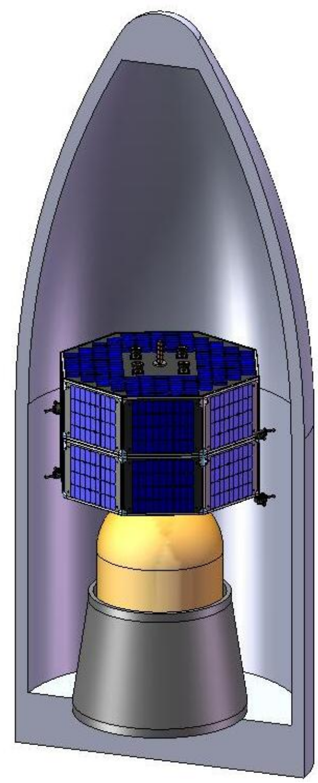

Figure 3: Blue Marble in Launch Configuration

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